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Two Stage Supersonic Inlet (TSSI): 10-inch Model Calculations

C.F. Smith and G.E. Smith
NYMA, Inc., Brook Park, Ohio

January 2005

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Glenn Research Center

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Acknowledgments

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Document History

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Note that at the time of writing, the NASA Lewis Research Center was undergoing a name change to the NASA John H. Glenn Research Center at Lewis Field. Both names may appear in this report.

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SUMMARY

NASA Lewis is currently engaged as a team member in the High Speed Research (HSR) Program. The HSR program is intended to provide technology readiness to the airframe and engine companies. The bifurcated inlet examined in this study (which is one of several being considered) was chosen based upon paper trade studies of axisymmetric, single sided, and bifurcated inlets. For a given compression ratio and mass flow a bifurcated inlet weighs less than a single sided inlet. An axisymmetric inlet has less bleed requirements than 2D inlets but has trouble matching transonic airflow requirements without going to a variable diameter centerbody. The bifurcated inlet was selected as one of the candidates because of its ability to match airflow schedules.

The inlet examined in this study, the Two Stage Supersonic Inlet (TSSI), was a candidate mixed compression bifurcated inlet for the HSR program. It has a novel concept to aid in inlet stability. Stability is the resistance of an inlet to disturbances causing unstart. A mixed-compression inlet has internal supersonic compression that can break down and lead to a condition known as unstart. For the TSSI, a large throat slot leading to a plenum with an exit area controlled by a fast acting valve was to be used to keep the inlet from unstarting. This large slot would conceptually function as a shock trap. A differential pressure sensor was used to open the plenum valve to maintain the plenum pressure at or slightly less than the pressure on the forward ramp surface. This concept was tested in the 10 x 10 wind tunnel at NASA Lewis.

CFD tools were used to predict and interpret the experimental results. 2D CFD calculations performed before the tests indicated that once the shock was located upstream of the aft slot lip, the inlet would unstart. The calculations also indicated that it was impossible to locate the shock at an arbitrary location within the slot. The shock was either at the slot's aft lip, further downstream or the inlet was unstarted. 3D calculations were initially computed at the pretest

design conditions and predetermined forward ramp angle of 12.32 degrees.

It was not possible to obtain a started solution. In the 10 x 10 wind tunnel at NASA Lewis, the experimentalists were able to obtain started data at the design geometry with the normal shock far downstream (highly supercritical condition). Also, they were able to vary the ramp angles to improve recovery and performance.

The test was performed from March to June 1994. The inlet had to be run at a forward ramp angle of 11.5 degrees to obtain critical performance data (normal shock near throat). As a check on the CFD results, a series of tunnel runs was completed at the pretest design condition's geometry and bleed flow rates. The test results verified the CFD calculations. To gain further insights into the workings of this inlet and to examine the three dimensionality of the flow it was decided to perform a 3D calculation for a supercritical case at ramp geometry and bleed flow rates that matched test data. The model configuration chosen had reasonable characteristics near the critical operating point.

The combination of a large number of 2D and a few 3D calculations was used to evaluate this inlet. It was necessary to perform the 3D calculation to examine the extent of flow nonuniformities, however the 2D calculations provided insights into the flow physics of this inlet. These results predicted the same phenomenon that occurred in the tests. It was very difficult to position the normal shock in a given position within the slot. If the back pressure was set too high or changed too abruptly the inlet would unstart. A large number of 2D calculations were performed in which the back pressure was gradually increased to move the normal shock forward. It would not have been possible to run such a large number of cases in 3D. At the Mach number and ramp angles used in this study the flow field within the inlet was two-dimensional for the most part except in the plenum region for the large slot where the flow exited the plenum sidewall.

The 3D calculations showed the flow in the 10" model to be two- dimensional with the exception of the flow in the plenum and a small corner vortex. Both the 2D and 3D calculations showed no benefit from the bleed chamber. It was not possible to locate the terminal shock at an arbitrary location within the slot region. Once the terminal shock was ahead of the aft lip it would start to travel upstream and the inlet would unstart. The sought after gain in stability from the large plenum was not realized in either the 10 x 10 test or the CFD results.

The computational results successfully predicted and reproduced two major problems encountered in the tests:

1. The inlet did not start at design ramp positions.
2. The slot does not enhance the stability of the inlet.

1.0 INTRODUCTION

The purpose of this study was to evaluate the TSSI inlet concept via the use of Computational Fluid Dynamics (CFD). Experimental tests were performed which were aimed more at obtaining performance data than obtaining flow details to evaluate the workings of the inlet. CFD calculations allowed a closer look at the details of the flow.

As a team member in the High Speed Research (HSR) Program, NASA Lewis was tasked to select an inlet for the next generation supersonic aircraft. A bifurcated inlet was chosen as one of the candidates based upon trade-off studies between single sided and bifurcated 2D inlets and axisymetric inlets. The bifurcated inlet in particular was chosen over the single sided inlet due to its shorter length and thus decreased weight and for its ability to match engine airflow schedules. The two-stage supersonic, TSSI, employs a novel concept developed by Koncsek[1]. Figure 1 shows a photograph of the inlet. A schematic diagram of the inlet is shown in Figure 2. The inlet is bifurcated so that the compression system is mirrored across the centerline. The shock wave from the ramp leading edge and the isentropic compression waves from the bend in the forward ramp coalesce and intersect at the leading edge of the cowl. The cowl shock wave intersects the area near the leading edge of the aft ramp. The design position is generally defined as the normal shock being located at the minimum geometric throat. In this inlet the aft ramp leading edge also defines the geometric throat. Theoretically, with the normal shock located at the throat, maximum inlet performance should be obtained. The throat Mach number is nominally 1.3. The bleed slot was intended to increase the stability by increasing the throat slot bleed flow thus slowing or stopping the forward movement of the shock. As the normal shock moves forward over the slot region the pressure increases in the plenum causing the increase in bleed flow. Also, the valves in the plenum would open to increase the choked area for the plenum sidewall bleed.

Boeing performed several tests under Independent Research and Development (IRAD) funds on a 2" wide model[2]. The inlet was configured to be controlled either manually or by a closed loop control system using the pressure difference between the forward ramp surface and the plenum to control the throat slot bleed exit mass flow rate. The 2" model tests provided promising results. The inlet appeared to be functional over a wide range of back pressures. Based on the success of the 2" model, a scaled up 10" wide model was designed and built for testing in the 10 x 10 tunnel at NASA Lewis Research Center. The supersonic diffuser for the 10" model was geometrically similar to the 2" model but the results obtained were not similar to the 2" model tests. An existing subsonic diffuser, provided by NASA lewis, was used for the 10" model. This diffuser was short, "with aggressive diffusion" the flow within the subsonic diffuser was separated which caused the inlet recovery to decrease. The 2" model diffuser was designed for the inlet and did not experience the separations and the resulting decrease in recovery.

CFD calculations were performed for an equivalent cross sectional area 2D geometry and for the 3D scale model. The transition region from a rectangular to a circular cross-section was modeled in 2D by adjusting the cowl wall distance from the ramp side wall to produce the same effective flow area as the three dimensional model. The CFD results complement the experimental data from the 10 x 10 wind tunnel. The use of CFD calculations allow the entire flowfield to be examined without the use of any intrusive measurement techniques.

2.0 TWO STAGE SUPERSONIC INLET DESIGN CONCEPT

The TSSI model, see Figure 1 and 2, has a large slot which opens into a plenum area. The bleed flow removed through the slot is controlled by a fast acting valve in the throat region of the plenum. The valve is controlled by a differential pressure sensor that attempts to keep the plenum pressure at the same value as the pressure on the ramp surface just ahead of the slot. If the sensor detects an increase in pressure it increases the plenum throat area and thus increase the mass removed by the plenum and decrease the plenum pressure. In practice the plenum pressure was kept slightly lower than the ramp pressure.

A boundary layer develops on the ramp surface and forms a shear layer across the bleed slot. Depending on the pressure in the plenum and the bleed mass flow rate, this shear layer can either be diverted into the plenum region and removed as plenum bleed or it could intersect the ramp wall on the downstream side of the slot. For the cases run in this study the flow was diverted into the slot. One would not want the flow to reattach since it would be more likely to separate than a freshly started boundary layer.

The shock from the leading edge of the ramp surface and the compression waves farther up the ramp coalesce and intersect the leading edge of the cowl. The cowl leading edge shock hits the leading edge of the aft lip or slightly downstream. A series of obliques reflecting alternatingly from the cowl surface to the ramp surface form a shock train within the inlet. As the back pressure is increased a normal shock forms at the end of the train and begins to move upstream. At the design condition the normal shock sits just off the lip, allowing the free shear layer flow from the forward ramp surface to jet into the plenum.

3.0 NUMERICAL MODELING

This section presents the code, NPARC, used in this study and a discussion of the boundary conditions and convergence criteria.

3.1 NPARC

NPARC version 1.1[3] was used for the 2D calculations. NPARC Version 2.0 was used for the 3D calculations. Both codes solve the Reynolds averaged Navier-Stokes equations in strong conservation form using the Beam and Warming approximate factorization scheme to obtain a block tridiagonal system of equations. Pulliam's scalar pentadiagonal transformation provides an efficient solver. The Baldwin Lomax Turbulence model[4] was used in the 2D calculations and the Baldwin-Barth Turbulence[5] model was used for the 3D calculation. A single grid block was used for the 2D calculations. The 3D code uses multiple grid blocks. Trilinear interpolation[6] transfers information at the grid block interfaces.

3.2 Boundary Conditions

3.2.1 Two-Dimensional Simulation

The inflow boundary condition was fixed at Mach 2.35 and 0 degrees angle-of-attack. The inlet back pressure was varied to position the shock at various locations within the inlet. Mass flow rate was specified for the cowl bleed surface. Plenum bleed was regulated by using a choked nozzle and varying the throat area. The bleed boundary condition adjusts the wall normal velocity to achieve a specified mass flow rate. The wall normal velocity is constant over the bleed region while pressure and density vary over the region.

3.2.2 Three-Dimensional Simulation

Inflow conditions were fixed at Mach 2.35. The exit pressure on the outflow boundary of the inlet was set to a very small value. This effectively located the normal shock at the exit plane. The exterior outflow boundaries were positioned so that the flow exiting them would be supersonic. On these surfaces, extrapolation was used. The porous bleed patches have specified mass flow rates. The plenum bleed was removed through an area on the sidewall of the plenum using fixed mass flow rate.

3.3 Grid Generation

3.3.1 2D Geometry

Figure 2 portrays the grid topology used in the 2D calculations of the Boeing TSSI model. Only the lower half of the inlet is shown. The plenum area is on top. For the 2D calculations, the size of the exit area in the plenum nozzle was varied to achieve two different bleed flow rates through the slot. The two areas corresponded to 10% and 30% percent of the inlet capture area. The removable lip is at the leading edge of the aft ramp. In the experimental study, three lip configurations were investigated. The sharp lip used in the pretest 2D calculations was not one of the lips tested. Figure 3 shows the 123 x 155 grid which were developed using the I3G[7] program. Several regions of the flow were blanked out using the patching capability of the PARC code. Viscous clustering was used on all the solid walls. There were typically 15 to 38 points within the boundary layers. Y plus values for the first point off the wall ranged from 0.5 to 3.0.

3.3.2 3D Geometry

Figure 4 shows a cross section of the grid used for the 3D calculation. A total of 10 grid blocks were used. Figure 5 shows the outlines of the 4 interior grid blocks. The initial grid, which contained a total of 1.5 million grid points was created by Sommerfield[1] of Boeing using Gridgen[8]. As in the 2D grid, viscous clustering was used on all the solid surfaces. The Y-plus values for the first point off the wall ranged from 1 to 5. The grid interfaces were continuous except for the overlapped grid in the slot region. The bleed zones were set up to remove boundary layer in regions where oblique and normal shocks were expected to occur. Figure 6 shows the bleed zones employed in the calculations. The stability bleed was removed through the sidewall on the rear portion of the plenum, labeled as "shock trap". The lip at the aft end of the bleed region is different than the sharp lip employed in the 2D calculations. During the experimental tests several lip configurations were examined. The lip that gave the best performance was used in the calculations. Figure 7 shows the lip detail of the fore and aft lips used for the calculations with the normal shock downstream of the throat.

3.4 Convergence Criteria

For the 2D calculations, it was possible to reduce the residuals by 10 or more orders of magnitude. Typically 30 to 50 thousand iterations were completed for each case. This number of iterations was possible due to the low computational requirements for this 2D problem. At this level of convergence there were no noticeable changes in the flowfield with additional iterations.

It was not possible to reduce the residuals for the 3D calculations by a similar amount. Depending on the block, the residuals were reduced from 5 to 10 orders of magnitude. The solution was run for 19,000 iterations, with iterations being carried out for all the blocks. There was a fluctuation in global bleed rate from 6.5% to 8.0%. This oscillation is attributed to the plenum chamber which has large regions of slow moving ($M < 0.1$), recirculating flow. The shock locations were fixed. A possible contributor to this plenum flow unsteadiness may be the use of a plane of symmetry boundary condition. In the actual tests, the plenum was open to both sides of the inlet and not divided as implied by the use of a plane of symmetry. The removal of this boundary condition and inclusion of the other side of the inlet may alleviate this unsteady flow situation. Although the inlet mass flow rate varied, the shock positions remain constant. However, as the bleed rate increased, the surface static pressures decreased.

4.0 RESULTS

4.1 2D Calculations

A matrix of calculations, in which the back pressure and plenum throat area were varied, was carried out on the TSSI model. Figure 8 shows pressure contours for a series of calculations in which the back pressure was increased causing the shock to move forward and the inlet to unstart. Note that there is no steady state solution with the shock located inside the slot region. Figure 9 shows a similar the same series of plots, but with a larger bleed plenum throat and hence more stability bleed. Under these circumstances the inlet was able to withstand an additional 10% increase in back pressure before unstart occurred. The plenum throat areas were 10% and 30% of the inlet capture area respectively. These calculations were performed on an equivalent 2D geometry. The calculations did not model the flow in a time accurate sense. In addition, the opening and/or closing of the throat in the bleed plenum was not modeled (which could be a key element in obtaining normal shock stability).

Viscous calculations were completed for supersonic flow throughout the inlet for a flight Mach number of 2.35 at an altitude of 20 km. The inlet was supposed to operate with the normal shock slightly ahead of the aft lip of the bleed slot. It was very difficult to locate the shock in that position. When the shock was downwind of the slot, the flow appeared to be separated along the ramp side of the inlet. As the back pressure was increased the shock moved upwind and the separation did not disappear until the shock was in the slot region. Once the shock was in the bleed region there was a very narrow range of back pressures over which it was stable. If the back pressure was increases beyond this range the shock would pop out, "unstart", unless the bleed was increased very quickly. In the 2 inch model experiment the bleed was increased by opening the doors in the plenum. The modeling did not include this.

To allow the terminal shock to remain in the vicinity of the throat region, it was necessary to bleed the cowl surface. The 2" model tests did not have cowl bleed, and boundary layer separation was suspected along the cowl due to the test recoveries not improving with increased bypass flow. The bleed was spread out over a length equal to 0.7 times half the cowl height and was centered under the terminal shock. Up to 3% of the capture mass flow as removed in a region directly under the aft lip. For most of the cases in which the shock foot rested on a solid surface, the flow downstream of the terminal shock was separated. For the higher plenum bleed cases the flow was not separated on the ramp side. The shock sits in an adverse pressure region and the shock strength was sufficient to cause separation. When the shock was located just off the lip, the cowl side flow remained attached. In this case, the boundary layer flow from the forward ramp did not reattach and instead was diverted into the plenum.

The design location for the shock was defined as being slightly ahead of the aft plenum lip. When the shock was in the design location, for both plenum throat areas, a perturbation in back pressure of the order of a few percent would cause the inlet to unstart. Increasing the plenum throat area from 10% to 30% and keeping the inlet back pressure constant caused the shock to be stable for an additional 10% increase in back pressure which brings the shock back to the design location. Once in the design location the shock was not stable to increases in the back pressure. Thus increasing the bleed in the plenum region is capable of keeping the inlet from unstarting due to decreases in engine mass flow if the actuators can react fast enough and if the large plenum slows down the movement of the shock.

In the 10" model wind tunnel tests it was generally not possible to prevent an unstart once the shock had "walked" into the slot region. Under certain circumstances a condition labeled as a "soft unstart" occurred. Under these conditions the inlet did not appear to be unstarted in the normal sense with the normal shock standing in front of the cowl surface. Surface static pressure measurements on the cowl would fluctuate with an increase in pressure levels indicating the possible presence of an unsteady normal shock. It was not possible to hold the shock at any arbitrary location within the slot region. It was only possible to hold it slightly off the aft lip. If the back pressure was increased further to move the shock further upstream the shock would travel upstream and keep going until the inlet was unstarted.

4.2 3D Calculations

It was not possible to obtain a started solution for the design ramp positions and lip geometry. The calculations were initiated with a restart file obtained from Boeing. After approximately 5000 additional iterations the solution would diverge. Extra bleed would remove the instability but it would return as soon as the bleed was removed. The flow would separate ahead of the fore lip on the inlet side of the plenum. The cause was believed to be from an instability developing within the plenum and slowly creeping around the fore lip. The grid used for this calculation had a slightly rounded fore lip. The grid used for the supercritical calculations had a sharp fore lip as shown in Figure 6. The inlet grid was revised with the ramp angles at the actual values used in the test and with the forward lip configuration used in the test. The percent of captured flow for cowl bleed was 2.26%, and for the side was 1.78%. The plenum bleed varied from 2.46% to 3.96% of captured flow.

Figure 10 shows the stagnation pressure contours at several locations within the inlet. Note a low pressure region in the cowl/sidewall corner. This represents a small vortex. Since the subsonic diffuser is not modelled, the effect of this vortex on the engine face total pressure distortion is not known. This vortex was also noted by Boeing in preliminary calculations. Another feature to note is that the ramp side boundary layer is much thinner behind the slot than ahead of it. This indicates that a significant portion of the boundary layer has been diverted into the slot, as expected.

Figure 11 shows Mach number planes at the symmetry plane, midway, and near the sidewall. Note a similar shock structure as in the 2D calculations. There is some small change in the shock structure near the wall due to boundary layer interactions and the corner vortex. Figure 12 shows a close-up of the Mach contours on the aft lip. Even though the shock reflects off the aft lip as an oblique it still does not contact the lip and appears to stand off it. Locally, very near the lip, the behavior is like that of a normal shock. The shock is normal to the surface and there is a jump in Mach number from supersonic to subsonic across it. The shock is, however, occurring very near a wall and flow is jetting perpendicular to the slip line into the bleed chamber in this region. The shock bends and becomes an oblique which reflects several times within the inlet duct.

Contours of turbulent viscosity are shown in Figure 13 at several locations within the inlet. The moderate levels of the viscosity contours and confinement to the boundary layer region indicates that the Baldwin-Barth turbulence model is functioning reasonably. Behind the slot, the ramp side turbulent viscosity levels are very low which is consistent with the removal of most of the upstream boundary layer by the slot. Another feature to note is the "roll-up" of turbulent viscosity contours in the inlet corner region at the entrance station. This is the same region that the corner vortex will develop downstream of the inlet entrance.

A comparison of surface centerline static pressures along the ramp and cowl are shown in Figure 14 for the supercritical case. Along the ramp, agreement is reasonable ahead of the slot. Behind the slot the data indicates a stronger shock farther upstream of the weaker one predicted by NPARC. Along the cowl, the numerical results also agree well with the data up to the end of the slot. Behind the slot, the numerical results predict a large shock impinging the cowl surface which is not indicated by the data. The surface static pressure levels downstream of the slot were reduced by up to 10% as the bleed flow rate increased from 6.5%

to 8% of capture flow rate.

A possible explanation for the significant discrepancies between the CFD results and experimental data behind the slot may be the presence of a normal shock in the experiment. The data indicates a similar pressure rise along the cowl and ramp at the same axial position. This implies the presence of a normal shock. This shock may be due to a large separation that may have occurred in the subsonic diffuser, which is not modeled in the numerical simulation.

5.0 CONCLUSIONS AND RECOMMENDATIONS

The calculations showed no unstart benefit of the slot bleed plenum. The computational results successfully predicted and reproduced two major problems encountered in the tests: 1. The inlet did not start at design ramp positions. 2. The slot does not enhance the stability of the inlet. Since the small scale tests indicated some measure of success, the differences between the two tests should be investigated. 2D calculations could be used to verify geometry changes such as the location of bleed bands and changes in ramp positions. A small scale test and/or 3D calculations could be employed if the 2D calculations show promise. This unsteadiness observed in the 3D calculations may not have been present in the 2D results and this difference may account for the different results. Employing baffles or other obstacles to dampen the vortical movements within the plenum chamber may reduce or eliminate the unsteadiness of the flow and provide a more stable shock capturing capability. The unsteadiness may be a numerical phenomena resulting from the use of a plane of symmetry in the plenum which was actually open to the other side of the inlet in the tests.

When the shock was in the design location and was disturbed by a pressure perturbation the inlet unstarted. The calculations have shown that an increased bleed is capable of holding the shock in place. If there is a benefit from the plenum it would be a transient phenomenon. To examine the buffering capability of the bleed chamber time accurate calculations would have to be performed.

Several new bleed models have been developed since the calculations were performed. The bleed boundary condition used on the plenum sidewall was probably adequate for the task, but the boundary condition for porous bleed was not capable of reflecting changes due to shock locations. Future calculations should make use of the advances made in bleed boundary conditions. The 3D calculations showed the flow in the 10" model to be relatively two-dimensional with the exception of the flow in the plenum and a small corner vortex. The 2D calculations were adequate to evaluate the operating conditions of this inlet.

REFERENCES

1. Koncsek, J.L., US Patent #4991795.
2. Mares, K.J., "Test Plan for 10 Biforced Two Stage Supersonic Inlet (BTSSI)", Boeing Document D6-57068.
3. Cooper, G.K. and Sirbaugh, J., "The PARC Distinction: A Practical Flow Solver", AIAA Paper No. 90-2002, July 1990.
4. Baldwin, B.S. and Lomax, H., "Thin Layer Approximation and Algebraic Turbulence Model for Separated Turbulent Flows", AIAA Paper 78-257, January 1978.
5. Baldwin, B.S. and Barth, T.J., "A One-Equation Turbulence Transport Model for High Reynolds Number Wall-Bounded Flows", NASA TM-102847.
6. Stokes, M.L. and Kneile, K.L., "A Search/Interpolation Algorithm for CFD Analysis", Presented at the World Congress on Computational Mechanics, University of Texas, Austin Texas, September 1986.
7. "I3G/VIRGO: Interactive Graphics for Geometry Definition/Visual Interactive Rapid Grid Generation, User's Manual", Wright Patterson AFB, January 1990.
8. Steinbrenner, J.P., Chawner, J.R., and Fouts, C.L., "The GRIDGEN 3-D Multiple Block Grid Generation System", WRDC- TR-90-30222, 1991.

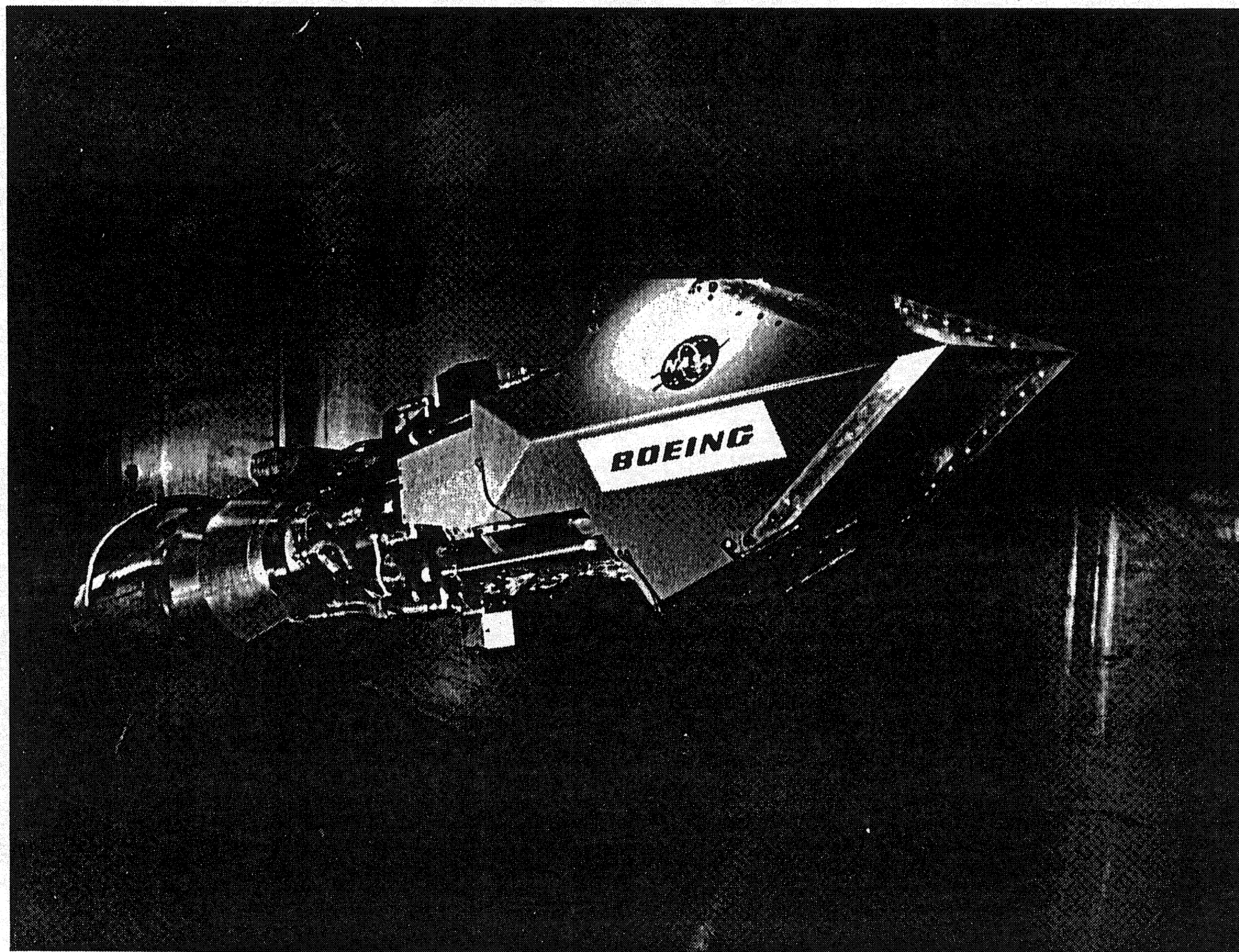


Figure 1 – Two Stage Supersonic Inlet (TSSI) 10" Model.

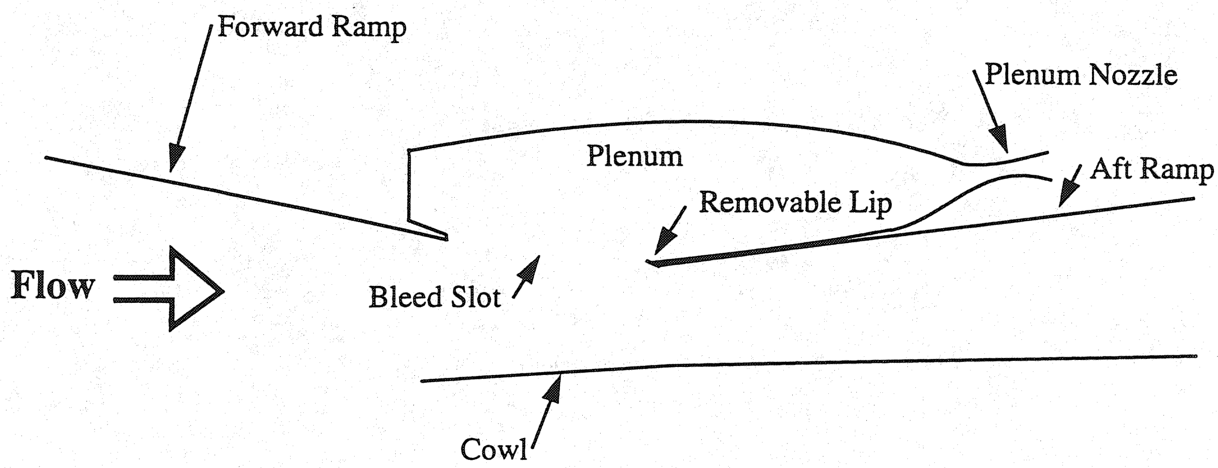


Figure 2 – Grid Topology Two Stage Supersonic Inlet (TSSI).

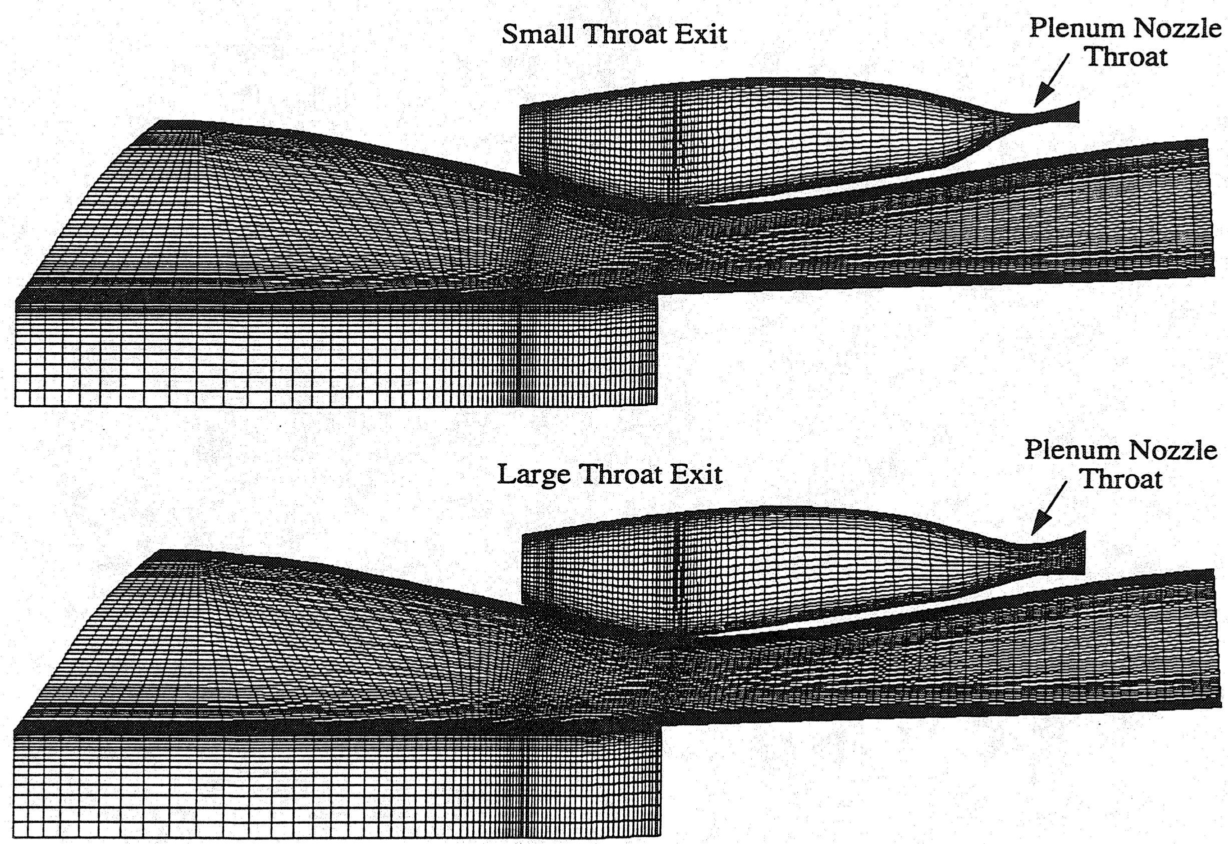


Figure 3 – Two–Dimensional Two Stage Supersonic Inlet (TSSI) Grid.

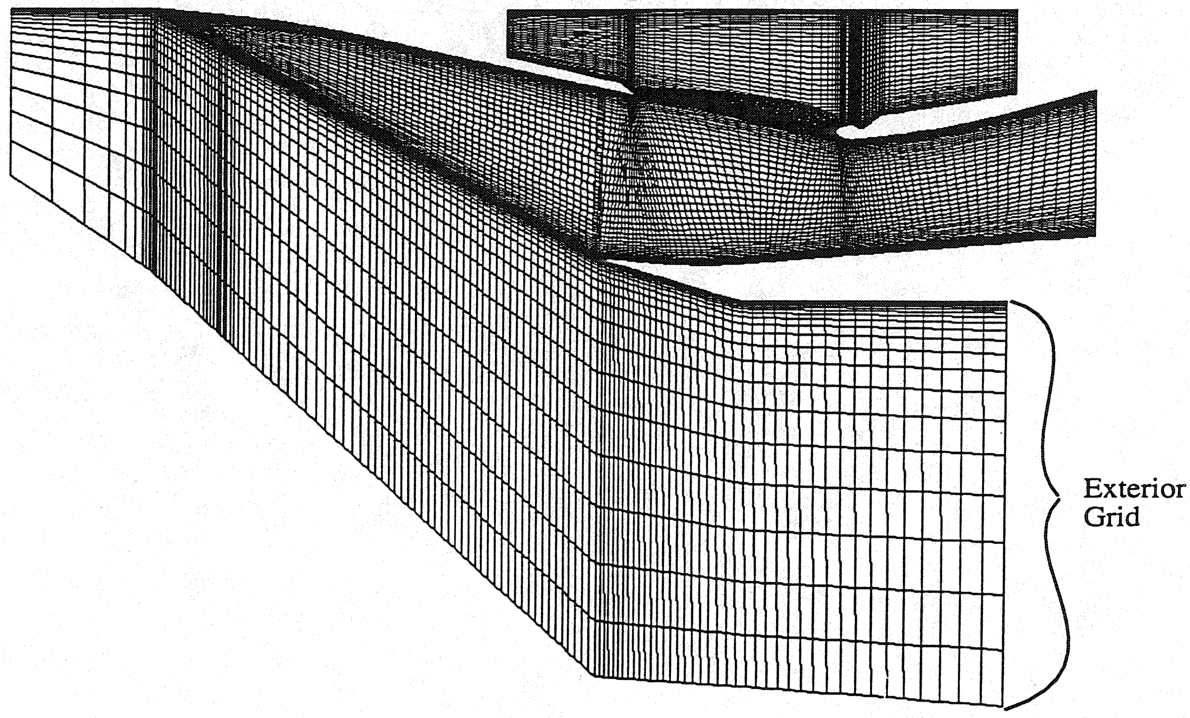


Figure 4 – Cross Section of Three–Dimensional Two Stage Supersonic Inlet (TSSI) Grid.

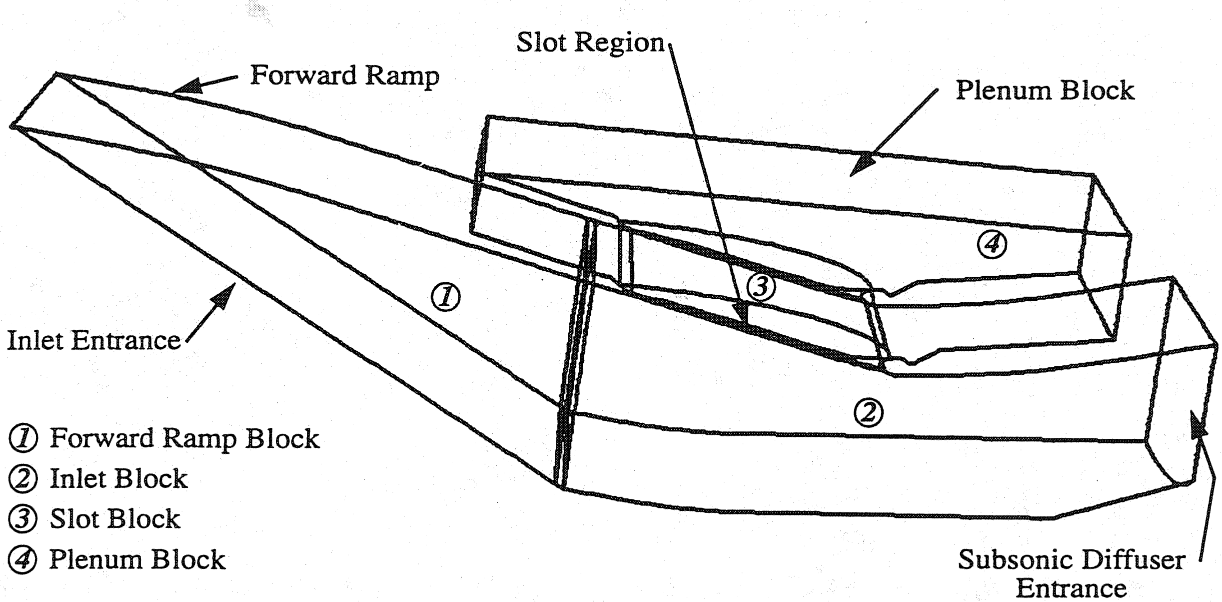


Figure 5 – Three–Dimensional Two Stage Supersonic Inlet (TSSI) Block Outline.

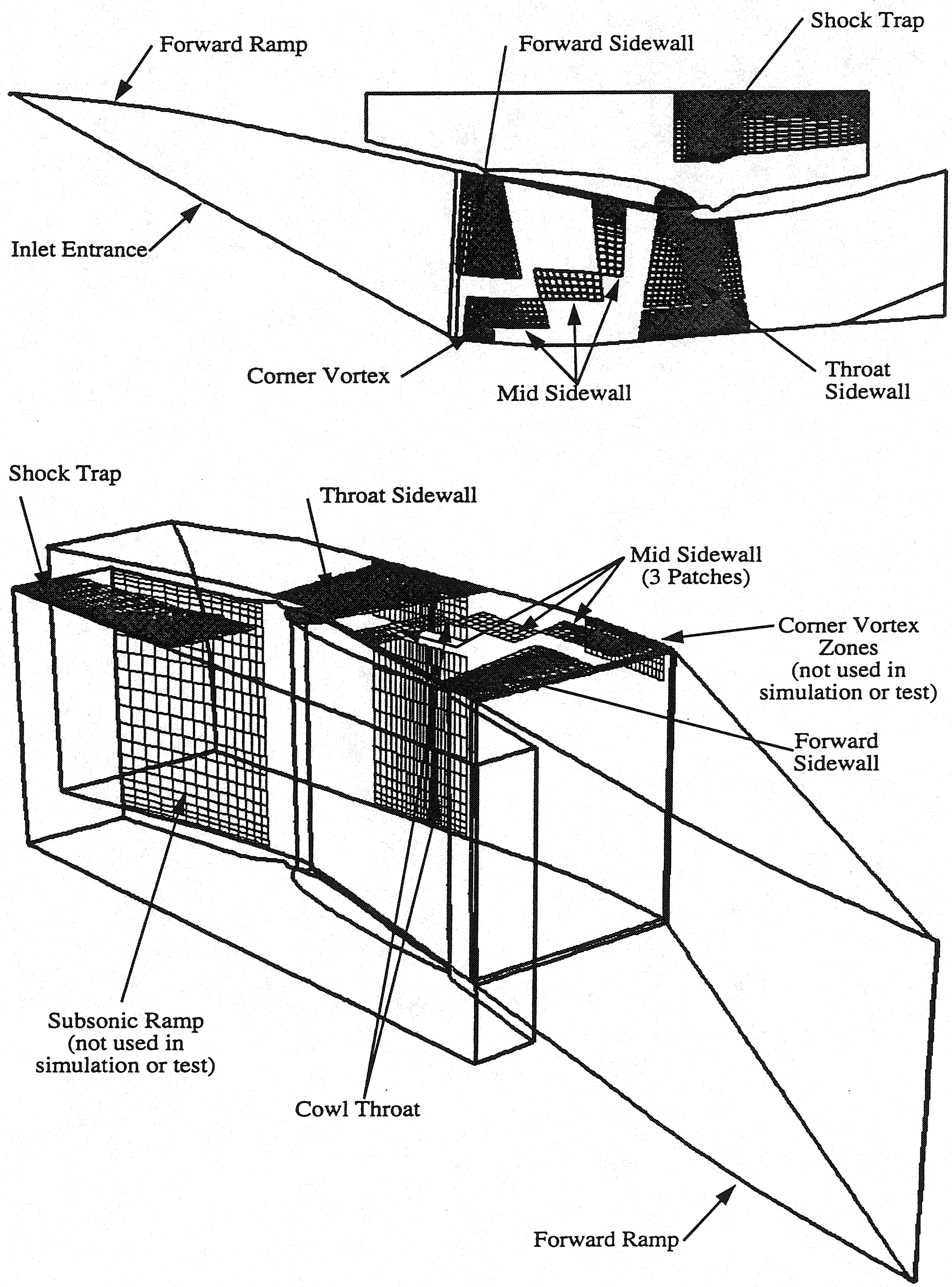


Figure 6 – Bleed Zones.

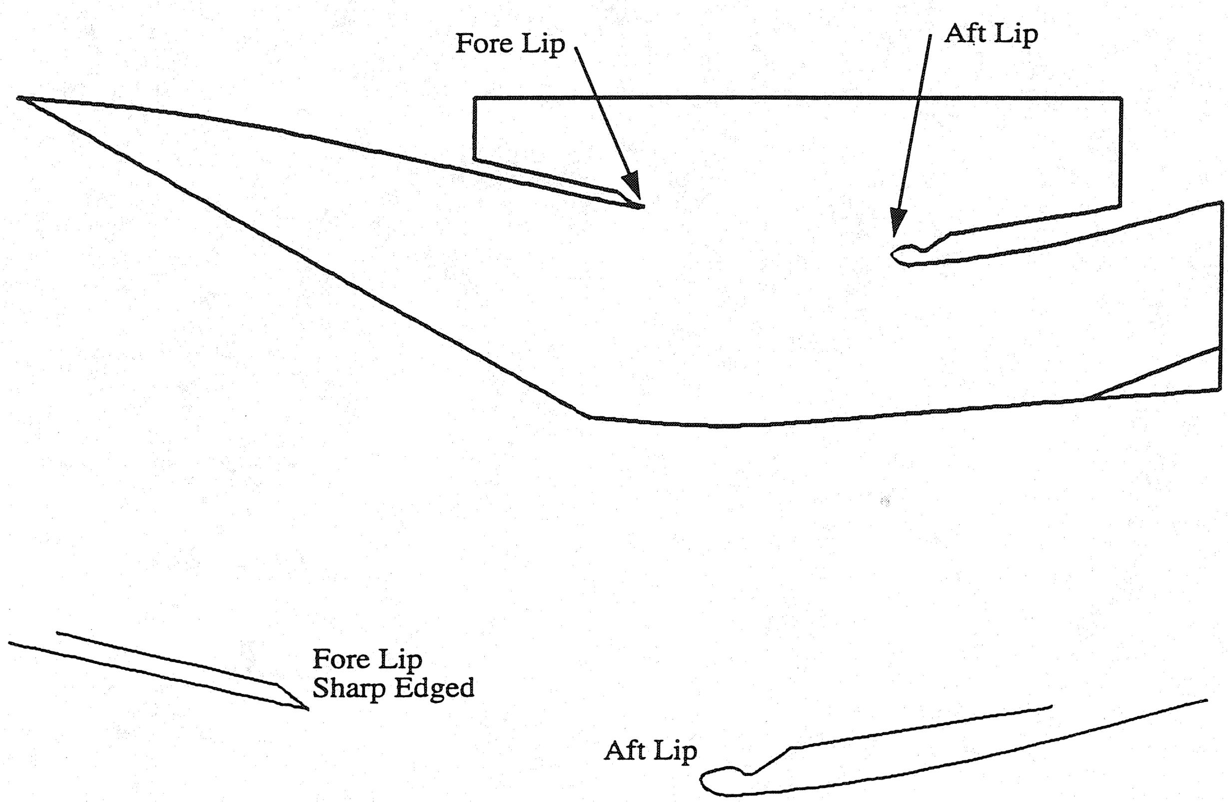


Figure 7 – Two Stage Supersonic Inlet 10" Model Lip Detail.

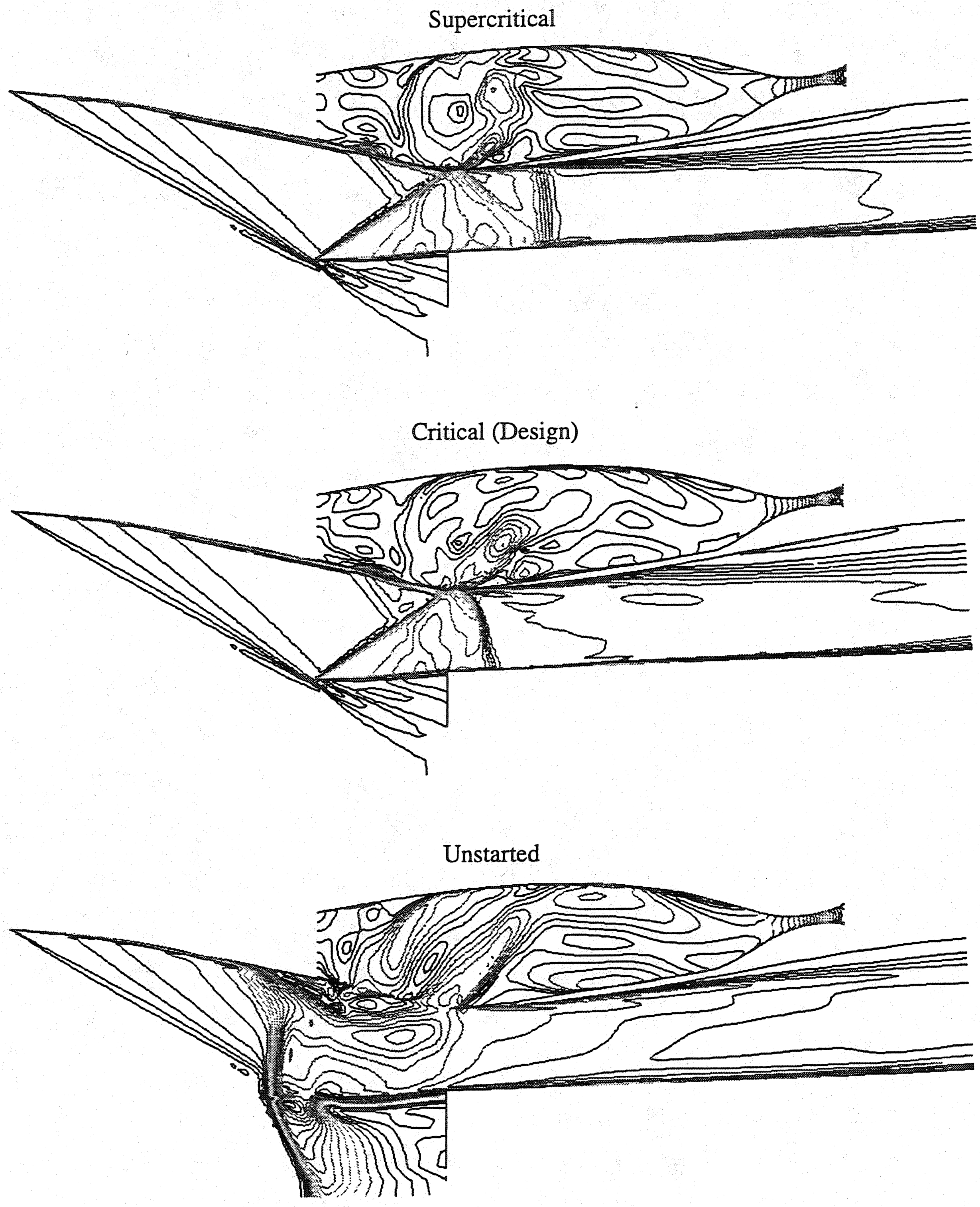


Figure 8 – Two–Dimensional Mach Contours (Small Nozzle Exit).

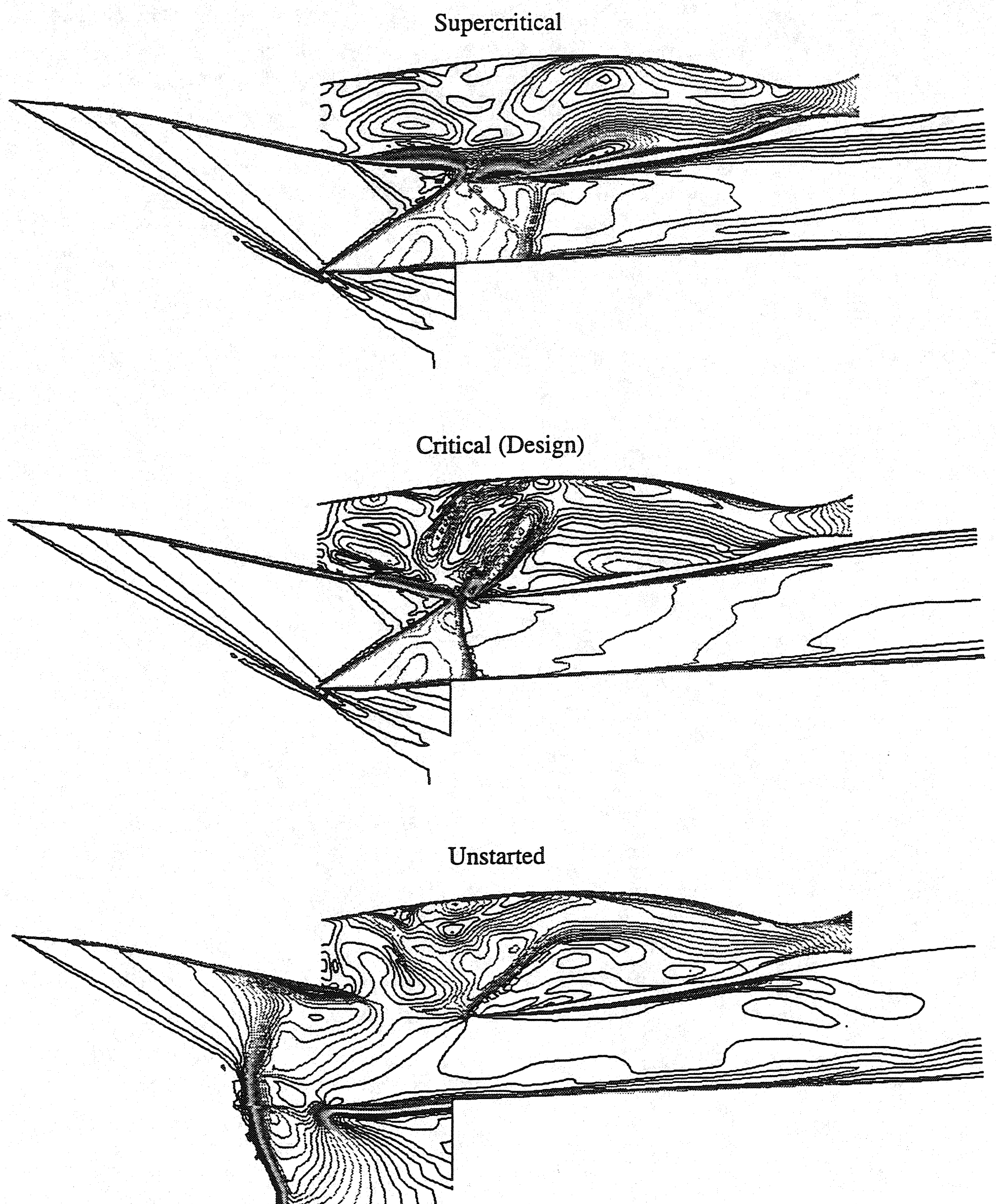
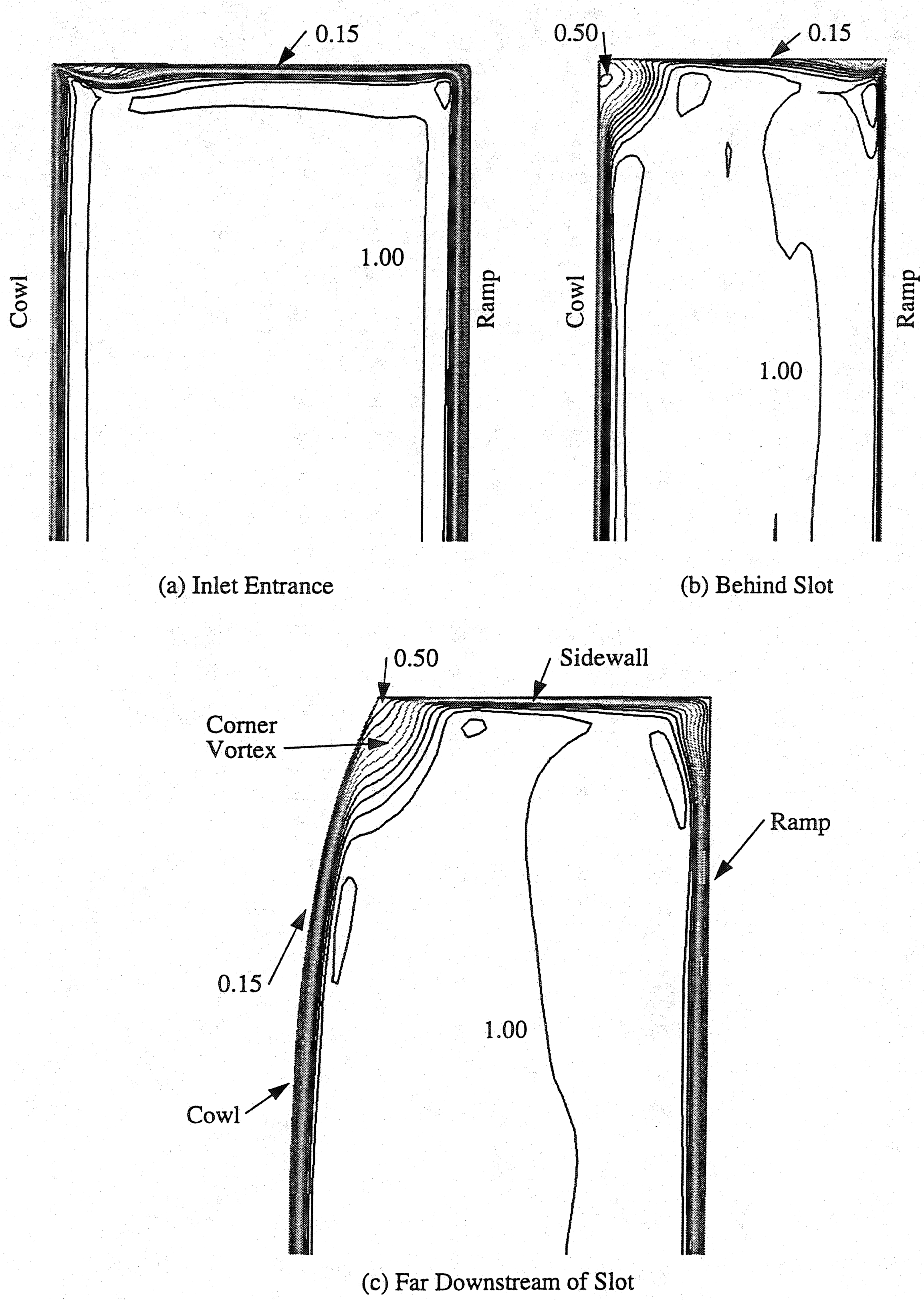
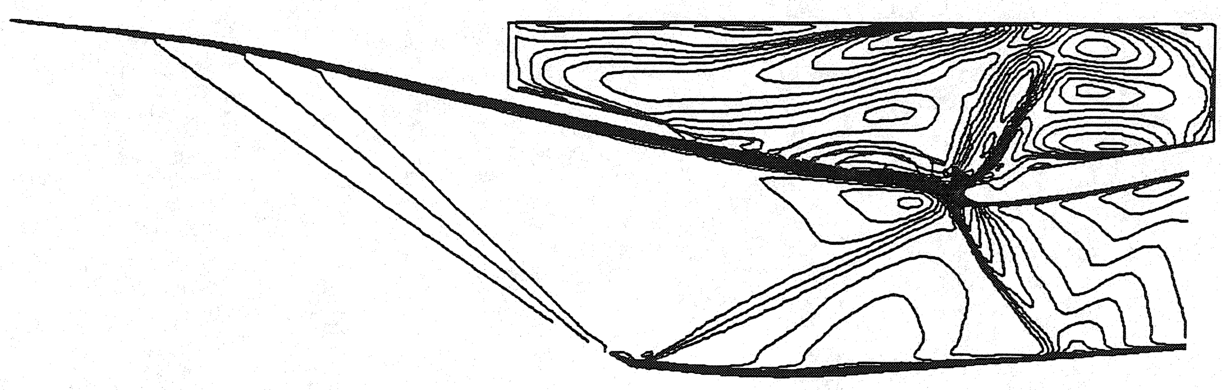


Figure 9 – Two–Dimensional Mach Contours (Large Nozzle Exit).

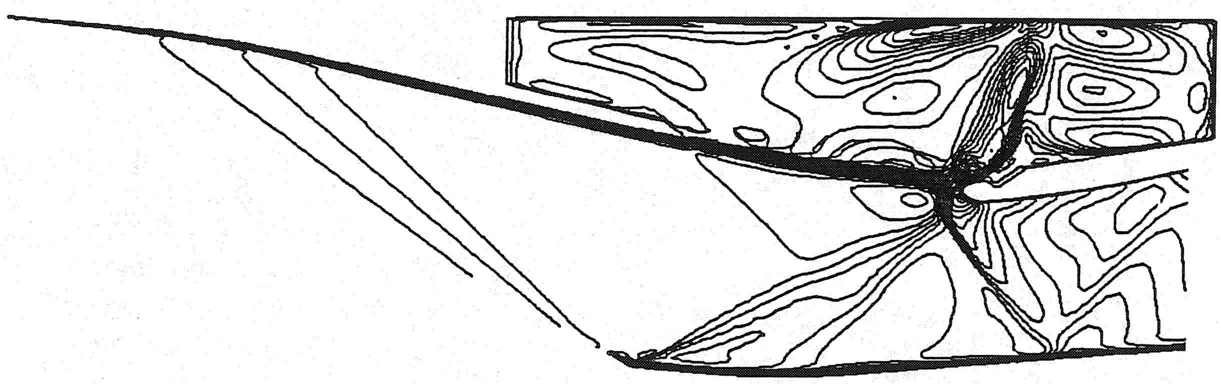


(c) Far Downstream of Slot

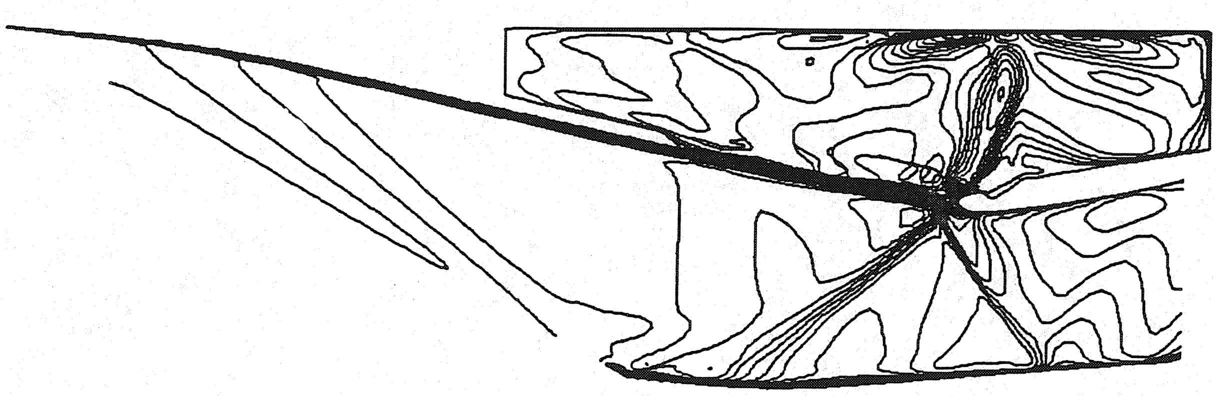
Figure 10 – Total Pressure Contours (P_T/P_{T_∞})



Symmetry Plane, Z = 0.0 inches.



Z = 1.67 inches.



Z = 3.014 inches.

Figure 11 – 3D Mach Contours.

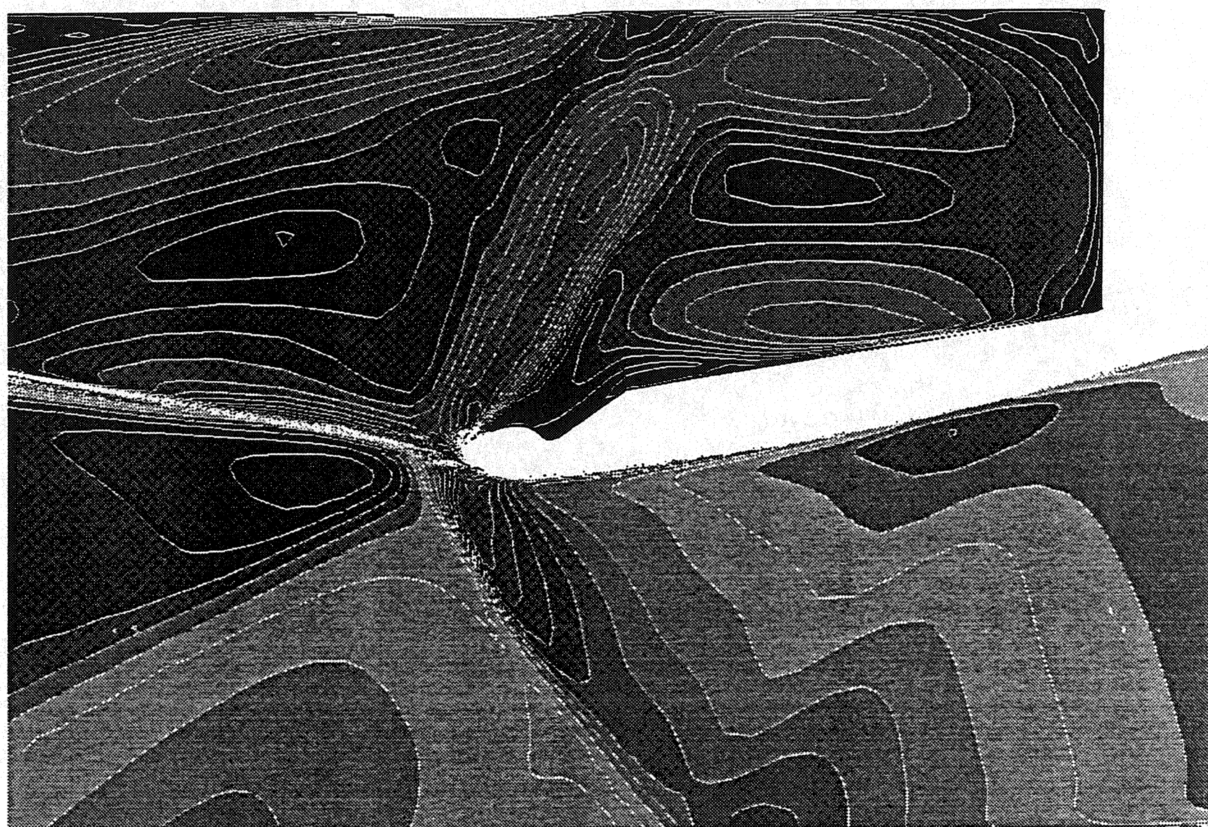


Figure 12 – 3D Mach Contour Close–Up Aft Lip.

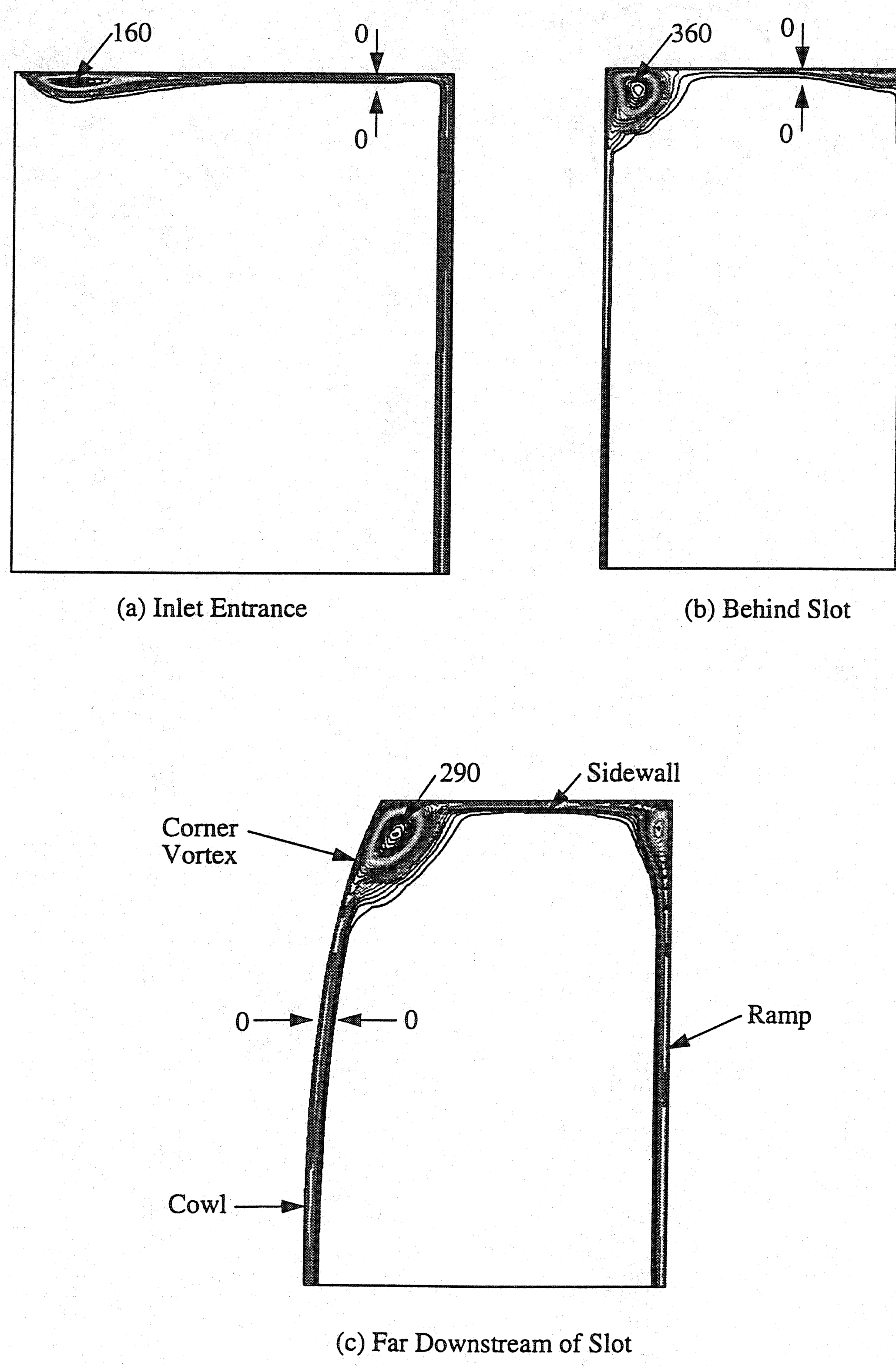
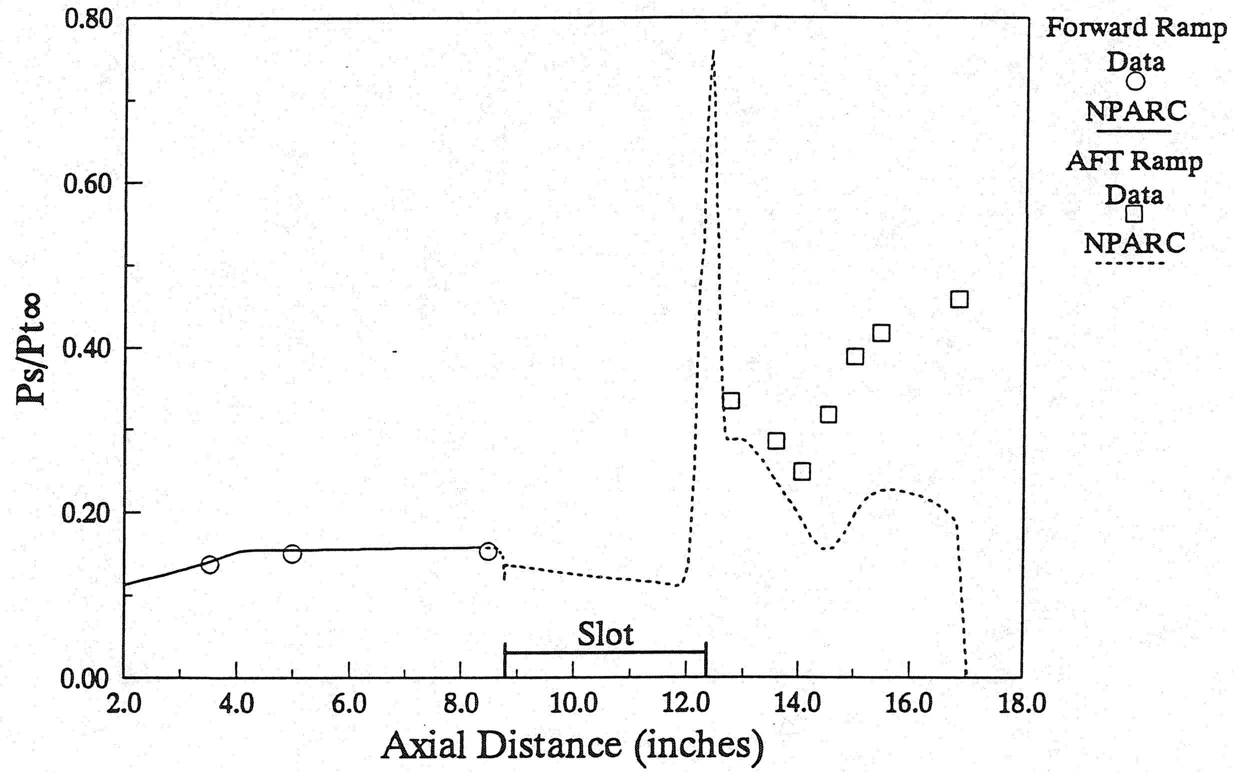
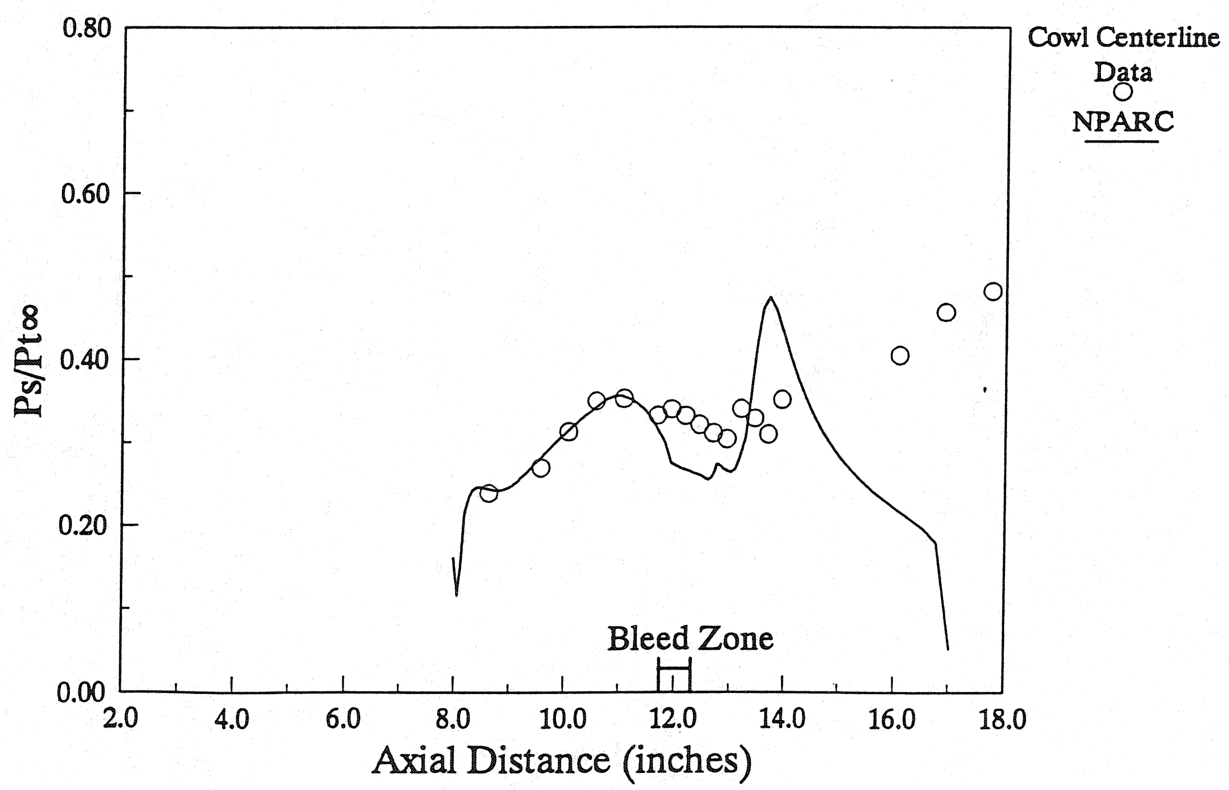


Figure 13 – Turbulent Viscosity Contours (μ_{Tur}/μ_{am}).



(a) Ramp Centerline



(b) Cowl Centerline

Figure 14 - 3D Surface Static Pressures.

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13. ABSTRACT ***(Maximum 200 words)***

The bifurcated inlet examined in this study (which is one of several being considered in the High Speed Research (HSR) Program) was chosen based upon paper trade studies of axisymmetric, single sided, and bifurcated inlets. For a given compression ratio and mass flow a bifurcated inlet weighs less than a single sided inlet. An axisymmetric inlet has less bleed requirements than 2–D inlets but has trouble matching transonic airflow requirements without going to a variable diameter centerbody. The bifurcated inlet was selected as one of the candidates because of its ability to match airflow schedules. The inlet examined in this study, the Two Stage Supersonic Inlet (TSSI), was a candidate mixed compression bifurcated inlet. It has a novel concept to aid in inlet stability. This concept was tested in the 10 × 10 wind tunnel at NASA Glenn. CFD tools were used to predict and interpret the experimental results.

14. SUBJECT TERMS

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